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APOLLO 8 MISSION REPORT

SERVICE PROPULSION SYSTEM FINAL FLIGHT EVALUATION

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
August 1970

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SERVICE PROPULSION SYSTEM FINAL FLIGHT EVALUATION

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SERVICE PROPULSION SYSTEM
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PURPOSE AND SCOPE

The purpose of this report is to present the results of the postflight analysis of the Service Propulsion System (SPS) performance during the Apollo 8 Mission. The primary objective of the analysis was to determine the steady-state performance of the SPS under the environmental conditions of actual space flight.

This report covers the additional analyses performed following the issuance of Reference 1, and the results reported herein supersede those contained in Reference 1 wherever differences exist.

Because this report is mainly concerned with the analysis of the SPS steady-state performance, little additional engine transient or thermal control analyses were performed beyond those reported in Reference 1.

The following items are the major additions to, or changes from, the results reported in Reference 1:

1) The performance values for the second SPS burn are revised.

2) The performance analysis of fourth burn was completed and the results are presented.

3) The analysis techniques, problems and assumptions are discussed.

4) The flight analysis results are compared to the preflight predicted performance.

5) The pressurization system performance is discussed in greater depth.

6) The transient data and performance for the third burn are included.

SUMMARY

The performance of CSM 103 Service Propulsion System during the Apollo 8 Mission was evaluated and found to be satisfactory.

A chamber pressure drop following ignition of the first SPS burn was attributed to a helium bubble trapped in the engine-oxidizer feed line because of an improper bleed during preflight servicing. The chamber pressure recovered prior to the end of the burn, and the three subsequent burns were normal.

The steady-state performance was determined by analyzing the second and fourth SPS burns using the Apollo Propulsion Analysis Program. The thrust during both these burns was less than predicted by approximately 2%, and was outside the expected -3 sigma limits. The less-than-predicted thrust resulted from propellant tank pressures which were less than expected. The decreased tank pressures were attributed to a change in the helium regulator outlet pressure resulting from a parts replacement prior to launch.

The engine performance corrected to standard inlet conditions for the second burn was as follows: thrust, 20441 pounds, specific impulse, 313.9 seconds, and propellant mixture ratio, 1.592. For the fourth burn the corresponding values were: 20465 pounds, 314.6 seconds, and 1.592, respectively. These values are less than 0.18% different from the values reported from the acceptance tests of the engine, and are well within the expected tolerances.

The oxidizer interface pressure measurement data was found to be erroneous during all burns, although appearing valid during coast. During the burns these data were biased approximately -8 psi.

The chamber pressure overshoot magnitudes during start for all four burns were noticeably decreased from those experienced on previous flights by starting in the single bore valve mode.

The Propellant Utilization and Gaging System was used for propellant loading, but was inactive during the flight. The PUGS was disconnected because of a suspected short circuit(s).

The SPS propellant line and engine valve temperatures were maintained well within their redline limits throughout the mission by passive thermal control. No SPS heater operation was required.

Based on the results of the flight analysis, the following recommendations are made:

1) Because of the somewhat unique nature of the erroneous oxidizer interface pressure measurement data, this instrumentation error should be investigated to preclude its recurrence on future flights should the error prove to be systematic.

2) The effects on predicted SPS performance of hardware changes, or adjustments, such as occurred with the helium regulator, should be assessed prior to launch to insure that the nominal, and 3 sigma, expected performance histories are still valid, and that there are no resulting performance effects detrimental to the mission.

3) The present methods of extrapolating the expected flight specific impulse from ground test data was satisfactory for this flight and need not be changed for future flights. This conclusion should be continually verified on each subsequent flight.

INTRODUCTION

The Apollo 8 Mission was the eighth in a series of flights using specification Apollo hardware, the second manned flight of a Block II spacecraft, and the first manned flight using a Saturn V launch vehicle. The mission was the first to the vicinity of the moon and was the continuation of a program to develop manned lunar landing capability. The overall objectives of the mission were to demonstrate Command and Service Module performance in a cislunar and lunar-orbit environment, to evaluate crew performance in a lunar-orbit mission, to demonstrate communications and tracking at lunar distances, and to return high-resolution photography of proposed Apollo landing areas and other locations of scientific interest.

Launch occurred at 7:51:00 am (E.S.T.) on 21 December 1968, from Kennedy Space Center (KSC), with the Apollo 8 spacecraft being initially placed into a parking orbit by the Saturn V launch vehicle AS-503.

After a parking-orbit coast period devoted to inflight systems checks, the third stage (S-IVB) of the launch vehicle was reignited at 2:50:37 GET for the translunar injection maneuver. This maneuver lasted for 319 seconds. At approximately 3:21:00 GET, the spacecraft was separated from the S-IVB using the service module reaction control system.

There were four SPS firings during the mission. The first SPS burn (MCC1) was a midcourse correction maneuver performed approximately 11 hours after liftoff which produced a velocity change of 20.4 ft/sec. Approximately 69 hours after liftoff the second SPS burn, the lunar orbit insertion burn (LOI-1), was accomplished. The resulting velocity change was 2997 ft/sec. Approximately 4 hours later a lunar orbit circularization maneuver was performed using the SPS. This was the third SPS burn (LOI-2) and required a 135 ft/sec velocity change. The fourth, and last, SPS burn (TEI) was the transearth injection maneuver which was performed approximately 89 hours 19 minutes after liftoff. The velocity change was 3517 ft/sec.

The actual ignition times and burn durations for the four SPS firings are shown in Table 1.

The Apollo 8 Mission utilized CSM 103 which was equipped with SPS Engine S/N 57 (Injector S/N 100). The engine configuration and expected performance characteristics (Reference 2) are contained in Table 2.

The Apollo 8 SPS configuration was very similar to the Apollo 7 configuration, which was the first flight Block II Apollo spacecraft. The major modifications to the SPS between Apollo 7 and Apollo 8 were the incorporation of the Mod 1E bi-propellant engine valve which will accommodate a lower temperature environment; the use of flow dividers in the zero-g retention reservoir to eliminate the propellant gaging system dynamic flow bias; and the deactivation of the flight combustion stability monitor.

The SPS engine was started in the single bore valve mode on all four burns to reduce the magnitude of the chamber pressure overshoot experienced on previous flights when starting in the dual bore mode. During the second (LOI-1) and fourth (TEI) burns the other bore was opened 3 to 4 seconds after ignition and the remainder of the burn was performed in the dual bore mode. The SPS PU valve was in the normal position throughout the mission.

The first three SPS maneuvers were no-ullage starts, while the fourth maneuver (TEI) was preceded by a 15 second +X SM RCS ullage maneuver to insure SPS propellant settling.

There was one Apollo 8 Mission Detailed Test Objective specifically related to the SPS. It was:

S3.21 SPS Evaluation

The functional test objectives of this DTO were:

1) Confirm the adequacy of the conversion of ground determined Isp to vacuum operation of the SPS.

2) Obtain data on the SPS performance for LOI and TEI burns.

3) Confirm the accuracy of the SPS propellant utilization and gaging subsystem in the auxiliary mode and compare the relative accuracy of the primary and auxiliary systems during the period of the burn when propellants are being depleted from the sump tanks.

4) Verify that the predictions of the thermal effects of a long duration SPS burn in a space environment are adequate for use in evaluating the design of the heat protection system.

The detailed requirements of this objective are described in Reference 3.

FIRST BURN HELIUM INGESTION

During the first SPS burn a momentary drop in chamber pressure occurred immediately following the initial chamber pressure buildup transient. As shown in Figure 1 , the duration of the pressure drop was 400 to 500 milliseconds, and the minimum pressure was approximately 51 psia. The drop was accompanied by low frequency oscillations, but the pressure recovered to the expected steady-state value and the remainder of the 2.4 second duration burn was normal. A coincident drop in oxidizer interface pressure was also observed, thus indicating that the cause of the chamber pressure drop was gas in the oxidizer feed line.

The presence of gas in the line is attributed to an inadequate engine-oxidizer bleed during preflight servicing which allowed a helium bubble to be trapped in the engine feed line. A review of the KSC propellant servicing procedures for CSM 103 revealed that an improper bleed procedure was used for the oxidizer feed system. A previous ground test in which the bleed procedure was also improperly conducted, thus leaving gas bubbles in the system, resulted in a similar chamber pressure trace. In addition, the second burn on AS-201 (S/C 009), which had helium ingestion, exhibited almost identical chamber pressure characteristics.

All of the trapped helium was apparently exhausted from the line during the first burn, and the chamber pressure histories for the subsequent burns were normal. Bleed procedures for future spacecraft have been revised to preclude the recurrence of trapped helium in the lines.

STEADY-STATE PERFORMANCE ANALYSIS

Analysis Technique

The major analysis effort for this report was concentrated on determining the flight steady-state performance of the SPS during the second (LOI-1) and fourth (TEI) burns. The first (MCC1) and third (LOI-2) burns were both of insufficient duration to warrant detailed performance analysis. The performance analysis was accomplished by use of the Apollo Propulsion Analysis Program which utilizes a minimum variance technique to "best" correlate the available flight and ground test data. The program embodies error models for the various flight and ground test data that are used as inputs, and by iterative methods arrives at estimations of the system performance history, propellant weights and spacecraft weight which "best" (minimum-variance sense) reconcile the availabe data.

Analysis Program Results

The Apollo Propulsion Analysis Program results presented in this report were based on simulations using data from the flight measurements listed in Table 3. The propellant densities were calculated from sample specific gravity data from KSC, flight temperatures of 70°F for the second burn and 71°F for the fourth burn, and flight interface pressures. The temperatures were estimated based on the data from all the feed-system temperature measurements. The estimated spacecraft damp weight (CSM minus SPS propellants) at ignition for both burns was obtained from the Apollo Spacecraft Program Office, and was assumed constant throughtout the burn. The initial estimates of the SPS propellants onboard at the beginning of the time segments analyzed were extrapolated from the loaded propellant weights.

Second Burn (LOI-1)

The SPS steady-state performance during the second burn was determined from the analysis of a 186-second segment of the burn. The segment of the burn analyzed commenced approximately 32 seconds after SPS ignition (FS-1), and included the flight time between 248932 and 249118 seconds G.E.T. The first 32 seconds of the burn were not included, to reduce any errors resulting from data filtering spans which include transient data, and because the acceleration data near the start of the burn exhibited trends which made it highly suspect. The time segment analyzed was terminated approximately 29 seconds prior to SPS shutdown (FS-2) for similar reasons.

The results of the Propulsion Analysis Program simulation of the second burn are contained in Table 4 along with the preflight predicted values. The values presented are for two time slices approximately 50 seconds, and 200 seconds following FS-1, and are considered representative of the actual flight values throughout the segment of burn analyzed. As shown in Table 4, the thrust and flowrates during the second burn were approximately 2% less than predicted. The less-than-expected thrust and flowrates resulted from the propellant tank ullage pressures being approximately 4 to 5 psi less than expected. The reasons for the decreased ullage pressures are discussed in the Pressurization System Evaluation section. The engine performance during this burn was satisfactory and was as would be expected for the decreased ullage pressures.

Fourth Burn (TEI)

The steady-state performance during the fourth burn was derived from the analysis of a 150-second segment of the burn. The segment analyzed began approximately 30 seconds following ignition (FS-1), and included the flight time between 321586 and 321736 seconds G.E.T.

The results of the Propulsion Analysis Program simulation of the fourth burn are presented in Table 5 along with the preflight predicted values. The values presented are for two time slices approximately 50 seconds, and 170 seconds, following FS-1, and are considered representative of the actual flight values throughout the segment of the burn analyzed. As observed for the second burn, the thrust and flowrates for the fourth burn were also less than predicted by approximately 2%. The cause of the reduced thrust and flowrates was again found to be less-than-anticipated propellant tank ullage pressures. The SPS performance during this burn was satisfactory. The thrust and flowrates during the time segment analyzed from the fourth burn were higher than the second burn values because crossover (storage tank depletion) occurred near ignition of the fourth burn. The increases in thrust and flowrates following crossover were close to the expected changes.

The Propulsion Analysis Program results for the second and fourth burns verified that data from the oxidizer interface pressure measurement (SP0931P) was erroneous. During all four SPS burns the data from this measurement was significantly lower than expected, even considering the lower propellant tank ullage pressures. For the second burn, the program computed that the measured data was biased -7.9 psi from the correct value. The program computed bias for the fourth burn was -8.4 psi, thus, verifying that the measured data was biased approximately -8 psi.

The erroneous oxidizer interface pressure does not, however, appear to be a simple measurement bias. During coast periods the interface pressure measurement data agreed very well with the oxidizer tank pressure data. It was only during the SPS burns that the data appears erroneous. The specified location for the interface pressure transducer is in a port upstream of the engine orifice and filter. Another pressure port, used during acceptance testing and normally plugged during flight, is located downstream of the engine orifice and filter. The pressure measured at this downstream port is typically about 8 to 10 psi less than that measured at the interface port. The S/N 57 engine was installed in the spacecraft at KSC, replacing the original engine allocated to CSM 103, thus, raising the possibility that the flight transducer could have been mislocated in the downstream port when the engine was changed. However, the engineering work sheets do not substantiate this theory, therefore, it cannot be verified. Because of the somewhat unique nature of this instrumentation error, it is recommended that it be further investigated to preclude its recurrence on future flights.

As observed on previous SPS flights, the measured chamber pressure appeared to exhibit a positive drift during both the second and fourth burns when compared to the program computed chamber pressure trends. The average magnitude of the apparent drift over the segments of the burn analyzed was approximately 0.01 psi/sec, being somewhat more pronounced near ignition on both burns. This drift is believed to result from thermal effects on the transducer. The time histories of the measured chamber pressure for the second and fourth burns are shown in Figures 2 and 3.

Good agreement existed, as seen in Tables 4 and 5, between the measured oxidizer tank, fuel tank and fuel interface pressures, and the comparable program computed values during both the second and fourth burns. The differences were generally 1 psi or less.

Critique of Analysis Results

Shown in Figures 4 through 15 are analysis program output plots which represent the residual errors, or differences between the filtered flight data and the program-calculated values. The figures represent thrust acceleration, chamber pressure, oxidizer interface pressure, fuel interface pressure, oxidizer tank pressure, and fuel tank pressure, in that order. The first set of residual plots is for the second burn analysis, and the second set is for the fourth burn analysis. The filtered flight data is also included on each plot.

A strong indication of the validity of the analysis program simulation can be obtained by comparing the thrust acceleration calculated in the simulation to that derived from the Apollo Command Module Computer (CMC) ΔV data transmitted via measurement CG0001V. Figures 4 and 10 show the thrust acceleration during the portion of the burns analyzed, as derived from the CMC data, and the residual error between the CMC and program calculated values. The residual error time histories have essentially zero means and little, if any, discernable trends. This indicates the simulations are relatively valid, although other factors must also be considered in critiquing the simulations.

Several significant problems were encountered in performing the steady-state analyses of the second and fourth burns which required that certain assumptions be made in order to obtain an acceptable match to both the flight and ground test data. These problems and the steps taken to resolve them are discussed below.

The acceleration data for the second burn exhibited several unexplained shifts and "humps", which according to the analysis program were not consistent with the other data. It was therefore necessary to edit-out certain segments of the acceleration data.

Both the second and fourth burn analyses were performed with an ullage pressure driven SPS model. This model utilizes filtered flight data from the measured tank pressures as the starting point for computing the pressures and flowrates throughout the system. The program is free to bias the tank pressures, if so required for a minimum variance fit, but the version used (Linear Model 0) is essentially constrained to follow the shape of the filtered tank pressure data. The results of initial simulations of the second burn yielded acceleration residual error profiles which did not have zero slopes, indicating a possible error in the thrust shape. Interface driven models gave poorer results. It was found that the addition of a small, positive, time correlated drift to the filtered tank pressure data resulted in a good acceleration match. The magnitude of the added drift was such that the pressure change over the entire time segment analyzed was less than 0.5 psia. Since the PCM one bit quantization for these measurements is approximately 1.0 psi, it is reasonable to assume that the filtered data may have shape errors of the magnitude in question.

The analysis of the fourth burn was complicated by a lack of confidence in the assumed chamber throat area versus burn time for this burn. Very little preflight data were available to characterize throat area changes for long duration burns which have been preceded by earlier long duration burns, such as the fourth burn following the 247 second LOI-1 burn, on this flight. Therefore, the initial simulation attempts were made assuming the same throat area profile as used in the preflight analysis (Reference 2). The preflight profile assumed an essentially constant area throughout the burn, and also assumed a small increase in area between the end of the second burn and start of the fourth burn due to cooling. It was found that by assuming a throat area decrease with burn time, similar to that used for the second burn, a good match to the data could be achieved. The resultant throat area profile, although different from the preflight assumed profile, appears quite reasonable when compared to the second burn profile. Figure 16 shows the preflight assumed throat areas for both the second and fourth burns, and the profile used in the fourth burn flight analysis. Because the confidence in the second burn preflight profile was much greater than in the fourth burn profile, no significant changes were made to it.

As previously discussed the measured chamber pressure appeared to exhibit a positive drift during both the second and fourth burns. Similar drifts have been consistently observed on previous flights. Because of the historical evidence that such a drift is characteristic of the transducer, an attempt was made to model the drift in the program. A constant drift, starting at ignition, with a rate of 0.01 psi/sec was assumed in the program computation of the measured chamber pressure, which is compared to the actual data from the chamberpressure measurement in the minimum variance match. Although some small trend errors still exist in the chamber pressure residuals (Figures 5 and 11) this model significantly improved the match on both burns.

It is recommended that this, or a similar, error model be used in subsequent flight analyses, with the assumed drift rate being updated as more flight data is accumulated.

Because the PUGS was inactive during the flight, it was not possible for the program to significantly decrease the propellant mass uncertainties from those assumed during loading. Therefore, the uncertainty in vehicle total mass was greater than with an active PUGS. However, the program did not indicate any unusually large vehicle mass errors during either burn, and it is felt that the assumed CSM dry mass and reported SPS propellant loads were accurate.

Comparison with Preflight Performance Prediction

Prior to the Apollo 8 Mission the expected performance of the SPS was presented in Reference 2. This performance prediction was for the integrated propellant feed/engine system and was characteristic of the SPS hardware on this flight. Thus, it was a preflight estimate of the propulsion system performance under space flight conditions, with no restrictions placed upon the conditions at the inlets to the engine.

The predicted steady-state thrust, specific impulse, and propellant mixture ratio for the second, third, and fourth burns are shown in Figure 17 versus the time from ignition for each burn. Also shown, for comparison, are the corresponding analysis program calculated flight performance histories for the portions of the second and fourth burns which were analyzed. As shown in Figure 17, and previously in Tables 4 and 5, the computed flight thrust is significantly less (approximately 1.9 to 2.3%) than the predicted thrust throughout both the second and fourth burns, and well outside the -3 sigma limits presented in Reference 2. The cause of the reduced thrust was, as previously noted, the less-than-predicted oxidizer and fuel tank ullage pressures. During both burns the ullage pressures were 4 to 6 psi less than expected. Based on a linearized engine model (influence coefficients) a 5 psi reduction in ullage pressure should, for example, result in a reduction in thrust of approximately 400 pounds. The less-than-predicted ullage pressures are attributed to the performance of the helium regulator and are discussed in the Pressurization System Evaluation section.

The analysis program calculated specific impulse and mixture ratio histories for the second and fourth burns are seen in Figure 17 and Tables 4 and 5 to agree with the predicted within the expected tolerances throughout the burn segments analyzed. The close agreement between the flight specific impulse and the predicted specific impulse, in spite of the large differences in tank pressure and thrust, demonstrates that, as predicted, the specific impulse is relatively constant over a rather broad range of tank pressures and thrust levels.

As discussed above, the flight thrust level for both the second and fourth burns was well outside the expected -3 sigma limits. This less-than-predicted thrust (and flowrates) was the major cause of these two burns being approximately 4.9 and 5.7 seconds longer, respectively, than planned to achieve the desired ΔV. (Reference 4). It should be emphasized that the nominal performance predicted in Reference 2 was based on certain specific characteristics for the CSM 103 SPS including, among others, the helium regulator outlet pressure characteristics based on the regulator acceptance test data. The uncertainties in the regulator outlet pressure (for a given inlet pressure) which were input to the prediction program to compute the SPS performance dispersions were therefore based on the assumption that the regulator set point was known. The assumed regulator outlet pressure uncertainty was ± 1.5 psia (3 sigma). As will be explained in the Pressurization System Evaluation section, it is believed that the set point of the regulator was altered, prior to launch, by a parts replacement at KSC. It appears that the regulator outlet pressure was decreased approximately 4 psi, or more, from the acceptance test value. This decrease is significantly greater than the -3 sigma value (-1.5 psi) used in the preflight dispersion analysis, which was intended only to account for statistical uncertainties, and not for hardware changes of this type.

It is highly recommended that whenever hardware changes, or adjustments, of this type are made that the Systems Analysis Section, Propulsion and Power Division (NASA/MSC) be promptly informed so that the possible effects on SPS performance may be assessed prior to launch. This will insure that any effects detrimental to the mission are identified, will allow the predicted performance to be revised where required, and will prevent the necessity of increasing the performance uncertainties to cover such situations.

Engine Performance at Standard Inlet Conditions

The expected flight performance of the SPS engine was based on the data obtained during the engine and injector acceptance test. In order to provide a common basis for comparing engine performance, the acceptance test performance is adjusted to standard inlet conditions. This allows actual engine performance variations to be separated from performance variations which are induced by feed system, pressurization system and propellant temperature variations.

The second burn engine flight performance, as determined by the analysis program, corrected to standard inlet conditions yielded a thrust of 20441 pounds, a specific impulse of 313.9 seconds, and a propellant mixture ratio of 1.592. These values are 0.05% greater, 0.06% less, and 0% different, respectively, than the values reported for the acceptance tests of the engine.

The analysis program results for the fourth burn, corrected to standard inlet conditions, yielded a thrust of 20465 pounds, a specific impulse of 314.6 seconds, and a propellant mixture ratio of 1.592. These values are 0.17% greater, 0.16% greater, and 0% different, respecitvely, than the values determined from the acceptance test data.

The standard inlet conditions performance values for the two burns agree well with each other. The small differences between the two burns, 24 pounds for thrust and 0.7 seconds for specific impulse, are within the tolerances of the analysis program results, and are not considered significant. The mean standard inlet conditions performance values for the two burns, therefore, were: thrust, 20453 pounds; specific impulse, 314.2 seconds; and propellant mixture ratio, 1.592. The standard inlet conditions performance values reported herein were calculated for the following conditions:

STANDARD INLET CONDITIONS

Oxidizer interface pressure, psia	162
Fuel interface pressure, psia	169
Oxidizer interface temperature, °F	70
Fuel interface temperature, °F	70
Oxidizer density, lbm/ft^3	90.15
Fuel density, lbm/ft^3	56.31
Thrust acceleration, lbf/lbm	1.0
Throat area (initial value), in^2	121.641

Of primary concern in the flight analysis of all Block II engines will be the verification of the present methods of extrapolating the specific impulse for the actual flight environment from data obtained during ground acceptance tests at sea level conditions. Since the SPS engine is not altitude tested during the acceptance tests, the expected specific impulse is calculated from the data obtained in the injector sea level acceptance tests using conversion factors determined from AEDC qualification testing. As previously discussed, the standard inlet conditions specific impulse values determined from analyses of the second and fourth burns were 313.9 seconds and 314.6 seconds, respectively, with a mean of 314.2 seconds. The predicted specific impulse at standard inlet conditions, as extrapolated from the ground test data was 314.1 seconds. The expected tolerances associated with this predicted value (Reference 2) were ± 1.59 seconds (3 sigma). The flight values for both burns, and the mean flight value, are well within these tolerances. Therefore, it is concluded that the present methods of extrapolating the expected flight specific impulse from the ground test data were satisfactory for this flight, and there is no evidence to warrant changing the methods for future flights. The validity of this conclusion should be continually verified on each subsequent flight.

Comparison of All Steady-State Data

The steady-state data for SPS burns 1 and 3 were also reviewed and compared to burns 2 and 4. Because these burns were relatively short, no detailed performance analysis was attempted. Table 6 contains the measured steady-state pressures for all four burns. The data indicate that the SPS steady-state performance was consistent during all four burns.

PUGS EVALUATION AND PROPELLANT LOADING

The Propellant Utilization and Gaging System (PUGS) was inactive for this mission. The PUGS was utilized for loading the vehicle, but during the prelaunch checkout the fuel sump tank primary probe and a fuel point sensor in the storage tank failed to operate properly, probably because of a short circuit. Because these malfunctions could not be readily corrected, and since there were relatively high SPS propellant reserves planned for this mission, it was decided to disconnect the PUGS by opening the PUGS circuit breaker. Since the circuit breaker is common to both the oxidizer and fuel gages, the oxidizer portion of the PUGS was also inactive during the flight. Therefore, no evaluation of PUGS operation during flight was possible.

The oxidizer tanks were loaded to a quantity readout of 100.9% at a tank pressure of 109 psia and an oxidizer temperature of 69°F. The fuel tanks were loaded to a quantity readout of 100.9% at a tank pressure of 113 psia and a fuel temperature of 70°F. A density determination was made at KSC from two oxidizer and two fuel samples of the SPS propellants. The analysis yielded an oxidizer density of 90.25% lbm/ft^3 at the loading temperature of 69°F and under a pressure of 109 psia. At 70°F and under a pressure of 113 psia, the fuel density was 56.51 lbm/ft^3.

Using these density values, and the above loading data, the SPS propellant loads were determined and are shown in the following table.

	Total Mass Loaded (lbm)	
Propellant	Actual[b]	Planned
Oxidizer[a]	25105	25090
Fuel[a]	15731	15695
Total[a]	40836	40785

[a]Includes gageable, ungageable, and vapor loaded quantities.

[b]Load reported by KSC in Spacecraft Operational Data Book.

PRESSURIZATION SYSTEM EVALUATION

Operation of the helium pressurization system was satisfactory without any indication of leakage. The helium supply pressure and the propellant ullage pressures indicated a nominal helium usage for the four SPS maneuvers.

The propellant tanks were pressurized to 179 psia for the oxidizer and 177 psia for the fuel four days prior to launch. The oxidizer tank pressure at liftoff was approximately 188 psia. The increase in ullage pressure prior to launch, especially the oxidizer tank pressure, is attributed to two factors; an increase in ullage temperature, and the resultant increase in propellant vapor pressure due to propellant surface temperature increase. The ullage temperature rise was caused by heat input from the fuel cell heaters located in the top of sector 4.

During the early portion of translunar coast (prior to the first burn), a drop of about 17 psi was noted in the service propulsion oxidizer tank pressure. The causes of this drop are believed to have been a decrease in ullage temperature and helium going into solution, with the pressure decrease stopping when the oxidizer became saturated. Both the ullage temperature decrease and the process of helium entering solution were accelerated by the zero "g" conditions during coast, which allow the propellants to migrate within the tanks, thereby greatly increasing the surface area. Because of this drop in pressure during coast, the oxidizer ullage pressure at ignition of the first burn was approximately 7 psi less than the expected value of 178 psia. The fuel tank ullage pressure decrease was less and the pressure was approximately the expected value of 178 psia at first burn ignition.

The measured steady-state propellant ullage pressures during the second and fourth maneuvers were less than expected - based on regulator acceptance test data - but, within the nominal ullage pressure limits of 178 $\pm$ 4 psia. As discussed previously, the decreased ullage pressures, which were 4 to 6 psi less than predicted, resulted in thrust and flowrates which were less than predicted. The controller section stems of the regulators which control ullage pressure were replaced at KSC prior to flight because of quality faults found in similar stems. Variations in manufacturing tolerances could account for the regulator pressures during flight being less than those recorded during acceptance testing, which was conducted using the original stems.

Immediately after cutoff of the second burn the oxidizer tank pressure was approximately 178 psia, having risen from the steady-state burn value of about 175 psia during the shutdown transient. During the coast period between the second and third burns the pressure increased approximately 11 psi to about 189 psia at third burn ignition. This increase is attributable to vapor resaturation and temperature recovery of the ullage. During the LOI-1 burn, the oxidizer ullage volume increased approximately from 11 ft^3 to 127 ft^3. The propellant in the storage tank was not completely expended during the LOI-1 burn, and there was sufficient propellant remaining at LOI termination to allow for complete saturation of the ullage with oxidizer vapor, thus, raising the ullage pressure by the partial pressure of the oxidizer.

ENGINE TRANSIENT ANALYSIS

A summary of the start and shutdown transients for the first, second, third and fourth SPS maneuvers is presented in Table 7. All transient data for the second, third, and fourth maneuvers were within specification limits. The transient times for the first maneuver were within specification limits. Both the start and shutdown impulse values for the first burn were less than their specification limits and outside their run-to-run tolerances with respect to the results of the second and fourth maneuver. However, in view of the less-than-nominal tank pressures and the helium ingestion experienced during the first maneuver, the transient impulse values are considered acceptable.

The engine was started in the single bore mode (valve Bank A) on all maneuvers, with a noticable decrease in the initial chamber pressure overshoot magnitudes, as compared to previous flights which utilized dual bore starts. During the second and fourth maneuvers the remaining bore (valve Bank B) was opened 3 to 4 seconds after ignition (causing an increase in chamber pressure as evident in Figures 2 and 3). This procedure has been accomplished numerous times during ground testing. The resulting system operation was nominal. The GN_2 actuation system pressures indicated nominal usage.

SPS THERMAL CONTROL

All service propulsion temperatures were maintained well within their redline limits. Passive thermal control, requiring roll rates of approximately one revolution per hour, was used during most of the translunar and transearth coast phases to maintain nearly stable onboard temperatures. This method of thermal control was interrupted only when specific vehicle attitudes were required. No SPS heater operation was required during the flight as the engine and system line temperatures, the oxidizer propellant utilization valve temperature, and the bipropellant valve temperature remained well within their limits with just passive thermal control. The minimum and maximum temperatures obtained during the flight at these locations are given in Table 8.

REFERENCES

1. TRW IOC 69.4354.2-8, "Apollo 8 Service Propulsion System Preliminary Flight Evaluation," from R. J. Smith to D. W. Vernon, dated 24 January 1969.

2. NASA Memorandum, "Mission C Prime - CSM 103 Service Propulsion System Preflight Analysis," J. G. Thibodaux, Jr., dated 17 October 1968.

3. SPD8-R-027, "Mission Requirements SA-503/CSM 103 C' Type Mission," dated 16 November 1968.

4. NASA/MSC Report MSC-PA-R-69-1, "Apollo 8 Mission Report," dated February 1969.

TABLE 1

SPS DUTY CYCLE

Burn	FS-1 (1)	FS-2 (1)	Burn Duration (secs)	Velocity Change (2) (ft/sec)
SPS 1	39599.19	39601.59	2.40	20.4
SPS 2	248900.40	249147.32	246.92	2997
SPS 3	264906.60	264916.21	9.61	134.8
SPS 4	321556.56	321760.29	203.73	3519

(1) Times are from Command Module Computer Downlink Data - CG0001V and are seconds G.E.T.

(2) Reference 4

TABLE 2

CSM 103 SPS ENGINE AND FEED SYSTEM CHARACTERISTICS

Engine No.	57
Injector No.	100
Chamber No.	334
Initial Chamber Throat Area (in.^{2})	121.6217
Oxidizer Engine Feedline Resistance (lb_f-sec^2/lb_m-ft^5)	497.
Fuel Engine Feedline Resistance (lb_f-sec^2/lb_m-ft^5)	882.
Oxidizer System Feedline Resistance (lb_f-sec^2/lb_m-ft^5)	97.72
Fuel System Feedline Resistance (lb_f-sec^2/lb_m-ft^5)	36.02

Characteristic Equation for C*:

$$C^* = C^*_{S.C.} + 870.5\,(MR - 1.6) - 273.83\,(MR^2 - 2.56) - 0.31878\,(P_C - 99) + 12.953\,(TP - 70) - 0.07414\,(TP^2 - 4900) - 5.466\,(MR \cdot TP - 112) + 0.03119\,(MR \cdot TP^2 - 7840.);$$ where $C^*_{S.C.}$ (Engine No. 57) = 5978.4 ft/sec

Characteristic Equation for I_{SP}:

$$I_{SP} = I_{SP_{vac}} - 96.954\,(1.6 - MR) - 0.0487\,(99 - P_C) - 0.06276\,(70 - TP) + 30.409\,(2.56 - MR^2) + 0.0004483\,(4900 - TP^2);$$ where $I_{SP_{vac}}$ (Engine No. 57) = 314.1 lb_f-sec/lb_m

TABLE 3

FLIGHT DATA USED IN STEADY STATE ANALYSIS

Measurement Number	Description	Range	Sample Rate Samples/Sec
SP0930 P	Pressure, Engine Fuel Interface	0 to 300 psia	10
SP0931 P	Pressure, Engine Oxidizer Interface	0 to 300 psia	10
SP0661 P	Pressure, Engine Chamber	0 to 150 psia	100
SP0003 P	Pressure, Oxidizer Tanks	0 to 250 psia	10
SP0006 P	Pressure, Fuel Tanks	0 to 250 psia	10
SP0048 T	Temperature, Engine Fuel Feed Line	0 to 200 °F	1
SP0049 T	Temperature, Engine Oxidizer Feed Line	0 to 200 °F	1
SP0054 T	Temperature, 1 Oxidizer Distribution Line	0 to 200 °F	1
SP0057 T	Temperature, 1 Fuel Distribution Line	0 to 200 °F	1
CG0001 V	Computer Digital Data	40 Bits	1/2

TABLE 4

SERVICE PROPULSION SYSTEM STEADY-STATE PERFORMANCE

SECOND SPS BURN

Parameter	FS-1 + 50 Seconds			FS-1 + 200 Seconds		
	Predicted	Measured	Calculated	Predicted	Measured	Calculated
Instrumented						
Oxidizer Tank Pressure, psia	179	174	174	180	175	175
Fuel Tank Pressure, psia	177	174	173	178	175	174
Oxidizer Interface Pressure, psia	166	154	161	166	154	162
Fuel Interface Pressure, psia	174	169	170	174	170	170
Engine Chamber Pressure, psia	102	100	100	102	102	100
Derived						
Oxidizer Flowrate, lbm/sec	40.9		39.9	40.8		39.9
Fuel Flowrate, lbm/sec	25.7		25.4	25.7		25.4
Propellant Mixture Ratio	1.59		1.57	1.58		1.57
Vacuum Specific Impulse, sec	314.2		313.9	314.2		313.9
Vacuum Thrust, lbf	20924		20491	20899		20488

NOTES:

(1) Predicted values from Reference 2

(2) Calculated values from Propulsion Analysis Program

TABLE 5

SERVICE PROPULSION SYSTEM STEADY-STATE PERFORMANCE

FOURTH SPS BURN

Parameter	FS-1 + 50 Seconds			FS-1 + 170 Seconds		
	Predicted	Measured	Calculated	Predicted	Measured	Calculated
Instrumented						
Oxidizer Tank Pressure, psia	181	175	175	181	176	176
Fuel Tank Pressure, psia	179	174	174	179	176	175
Oxidizer Interface Pressure, psia	171	158	166	171	158	166
Fuel Interface Pressure, psia	177	173	173	178	173	174
Engine Chamber Pressure, psia	104	102	102	104	104	102
Derived						
Oxidizer Flowrate, lbm/sec	41.7		40.7	41.7		40.6
Fuel Flowrate, lbm/sec	26.0		25.5	26.1		25.6
Propellant Mixture Ratio	1.60		1.59	1.60		1.59
Vacuum Specific Impulse, sec	314.3		314.7	314.3		314.7
Vacuum Thrust, lbf	21287		20830	21297		20854

NOTES:

(1) Predicted values from Reference 2

(2) Calculated values from Propulsion Analysis Program

TABLE 6

SUMMARY OF MEASURED STEADY-STATE PRESSURE DATA

MEASUREMENT	BURN					
	1	2		3	4	
	FS1 + 2 sec	FS1 + 50 sec	FS1 + 200 sec	FS1 + 7 sec	FS1 + 50 sec	FS1 + 170 sec
SP0003P Oxidizer Tank	169	174	175	180	175	176
SP0931P Oxidizer Interface[1]	152	154	154	159	158	158
SP0006P Fuel Tank	169	174	175	172	174	176
SP0930P Fuel Interface	167	169	170	175	173	173
SP0661P Chamber	95	100	102	96	104	104

NOTE: All values in psia.

[1] Measured data considered erroneous. Data biased by approximately -8 psi

TABLE 7
ENGINE TRANSIENT DATA

Parameter	Specification Value		Apollo 8 S. . Maneuvers			
	Single Bore	Dual Bore	1st	2nd	3rd	4th
Total Vacuum Impulse (Ignition to 90% Steady-State Thrust), lbf-sec	450 ± 250 (a) ± 200 (b)	-	444.5	504.7	665.5	683.7
Time (Ignition to 90% Steady-State Thrust), sec	0.675 ± 0.100	-	0.674	0.660	0.620	0.650
Chamber Pressure Overshoot, Percent	120	-	-	113	110.5	112.8
Total Vacuum Impulse (Cutoff to 0% Steady-State Thrust), lbf-sec	12,500 ± 2,500 (a) ± 500 (b)	13,500 ± 2,500 (a) ± 500 (b)	8,509.2	11,625	10,031	11,933
Time (Cutoff to 10% Steady-State Thrust), sec	1.075 ± 0.175	1.075 ± 0.175	0.934	1.24	0.901	1.01

(a) Engine to Engine Tolerance
(b) Run to Run Tolerance

NOTE: Maneuvers 1 and 3 had single bore starts and shutdowns.
Maneuvers 2 and 4 had single bore starts and dual bore shutdowns.

TABLE 8

SPS TEMPERATURES

Measurement Number	Measurement Description	Temperature (°F)			
		Minimum		Maximum	
		Redline	Actual	Redline	Actual
SP0045T	Engine Bipropellant Valve	30	62	160	121*
SP0048T	Fuel Engine Line	25	64	110	85
SP0049T	Oxidizer Engine Line	25	65	110	85
SP0054T	Oxidizer System Line	30	62	110	71
SP0057T	Fuel System Line	30	67	110	84
SP0617T	Oxidizer PU Valve Inlet	30	61	110	83
SP0618T	Oxidizer PU Valve Outlet	30	59	110	79

*Maximum soak back temperature occurring after SPS transearth injection maneuver.

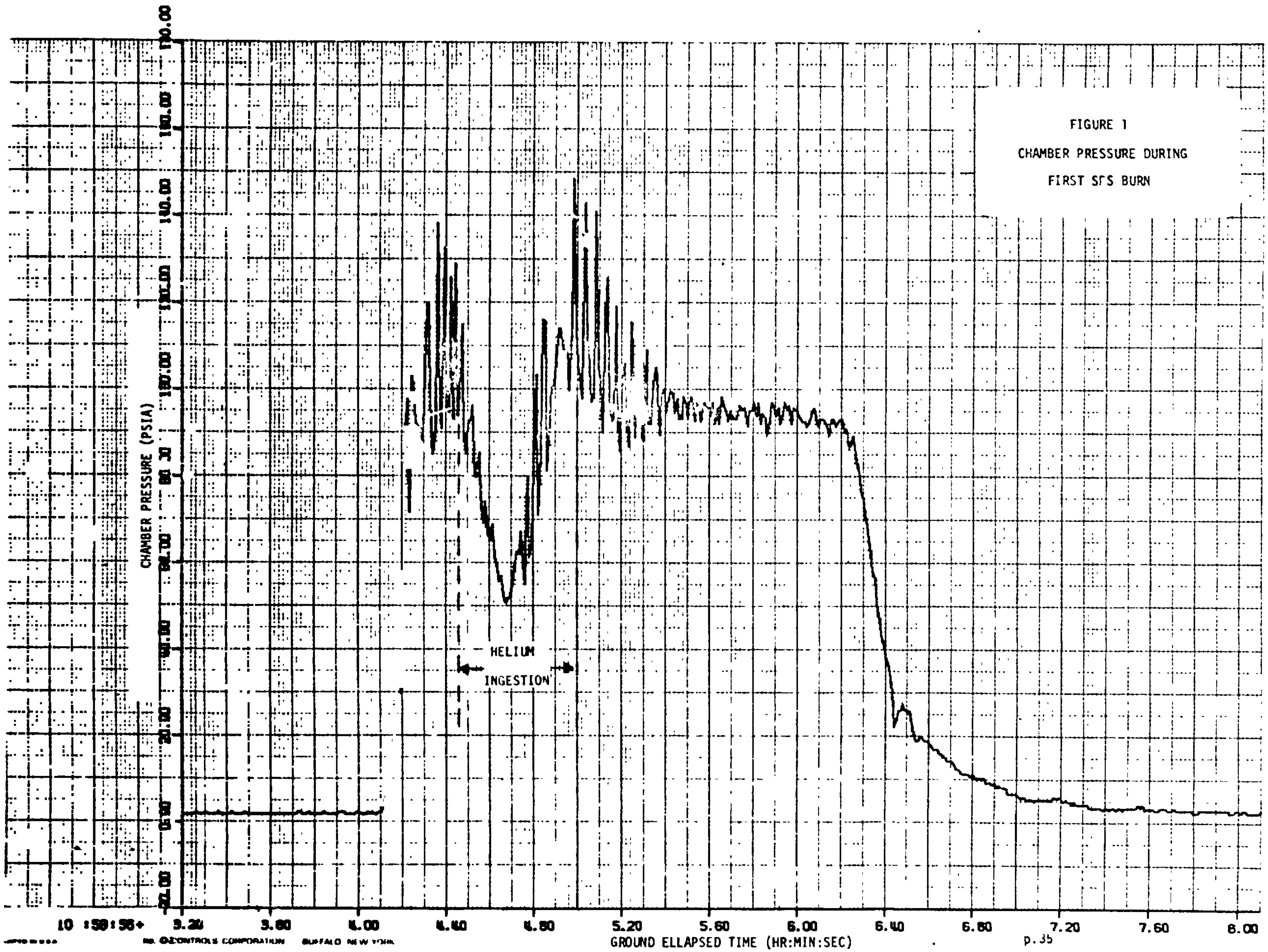
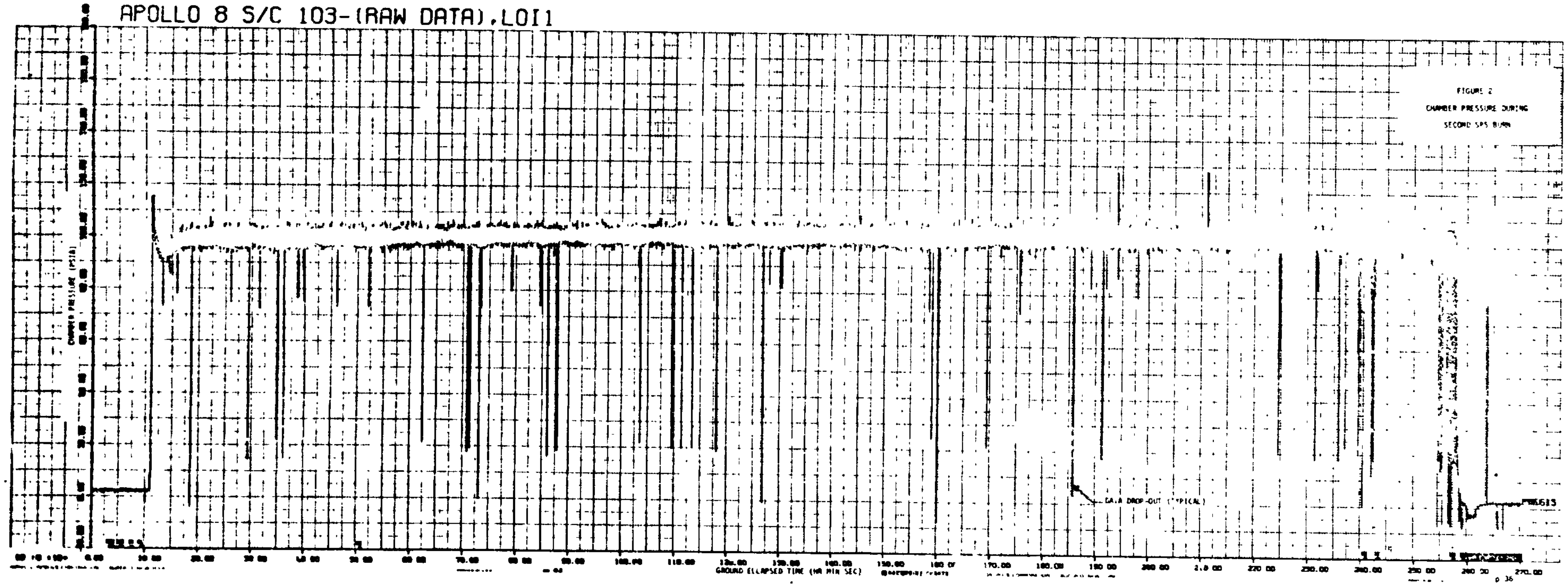
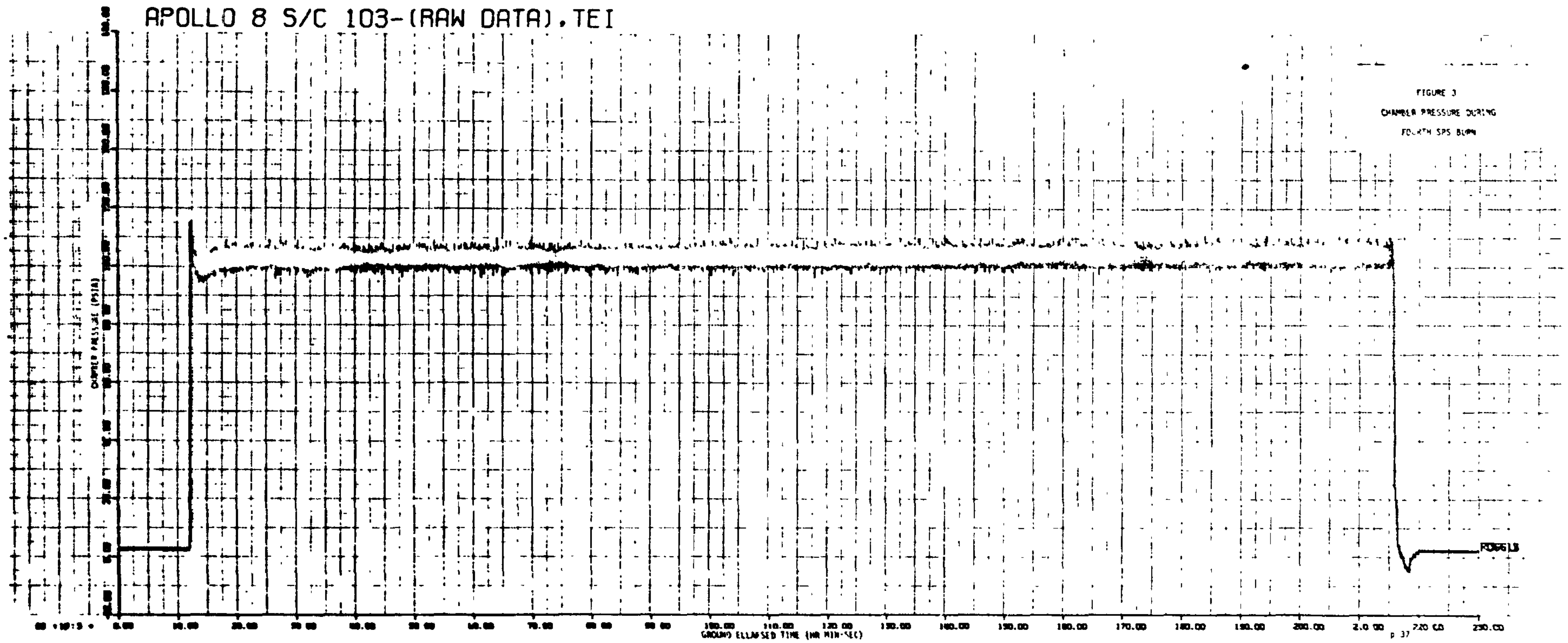


FIGURE 1
CHAMBER PRESSURE DURING
FIRST SPS BURN
HELIUM
INGESTION
CHAMBER PRESSURE (PSIA)
GROUND ELLAPSED TIME (HR:MIN:SEC)
10 :58:56+
3.20
3.60
4.00
4.40
4.80
5.20
5.60
6.00
6.40
6.80
7.20
7.60
8.00


APOLLO 8 S/C 103-(RAW DATA),LOI1
FIGURE 2
CHAMBER PRESSURE DURING
SECOND SPS BURN
CHAMBER PRESSURE (PSIA)
GROUND ELLAPSED TIME (HR MIN SEC)
DATA DROP-OUT (TYPICAL)


APOLLO 8 S/C 103-(RAW DATA), TEI
FIGURE 3
CHAMBER PRESSURE DURING
FOURTH SPS BURN
CHAMBER PRESSURE (PSIA)
GROUND ELAPSED TIME (HR MIN SEC)
P0661

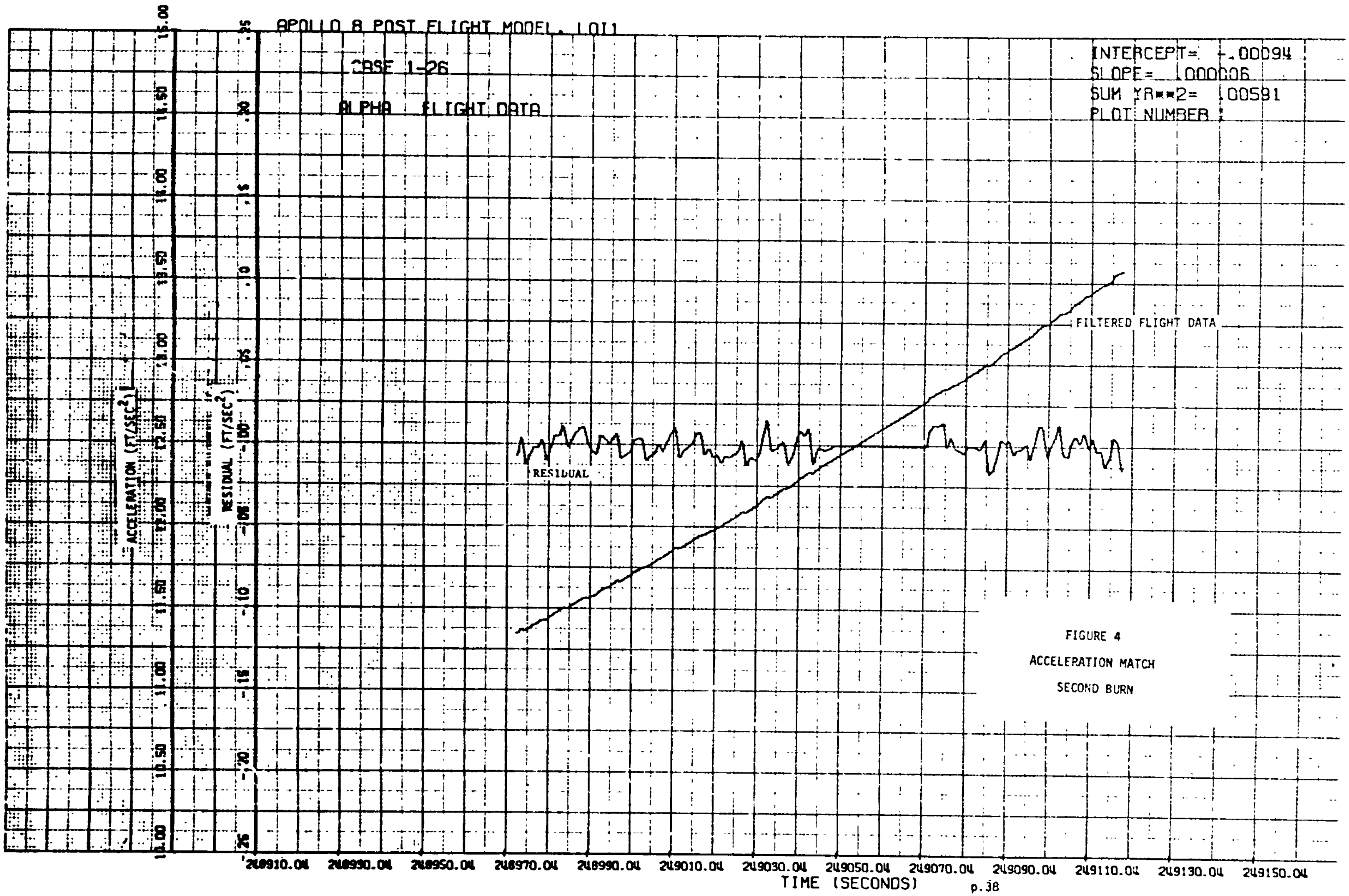


FIGURE 4
ACCELERATION MATCH
SECOND BURN

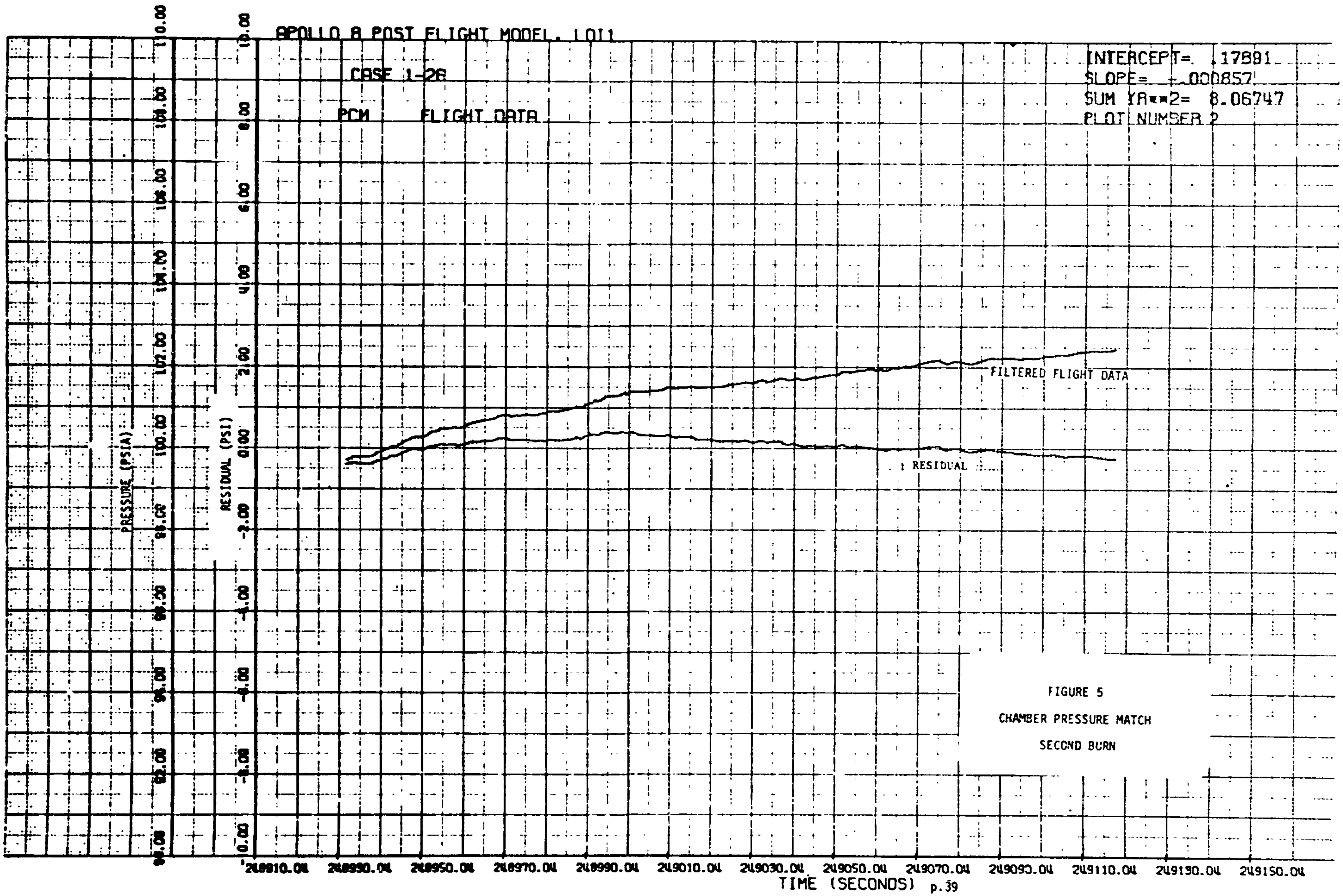


FIGURE 5
CHAMBER PRESSURE MATCH
SECOND BURN

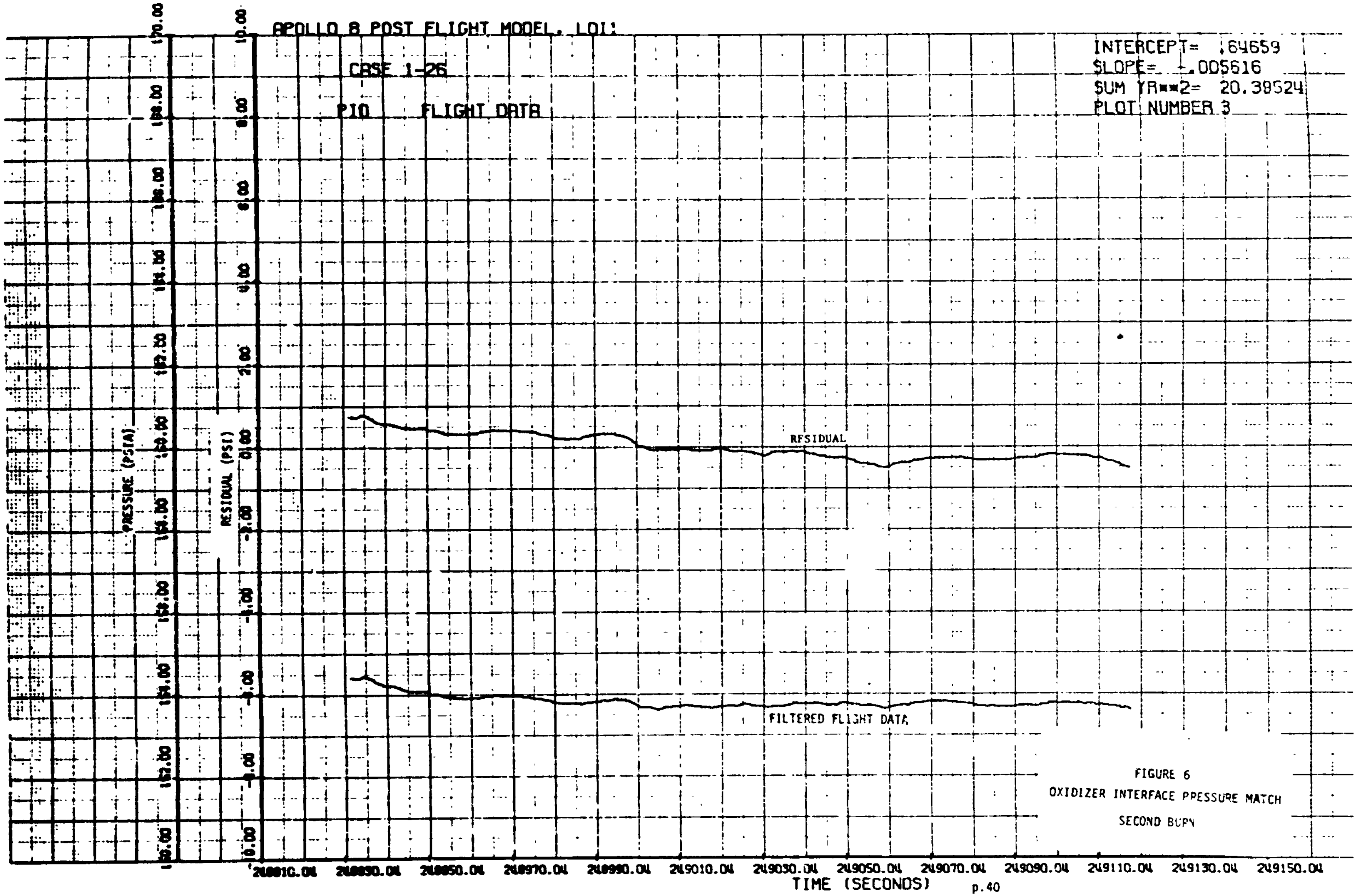


FIGURE 6
OXIDIZER INTERFACE PRESSURE MATCH
SECOND BURN

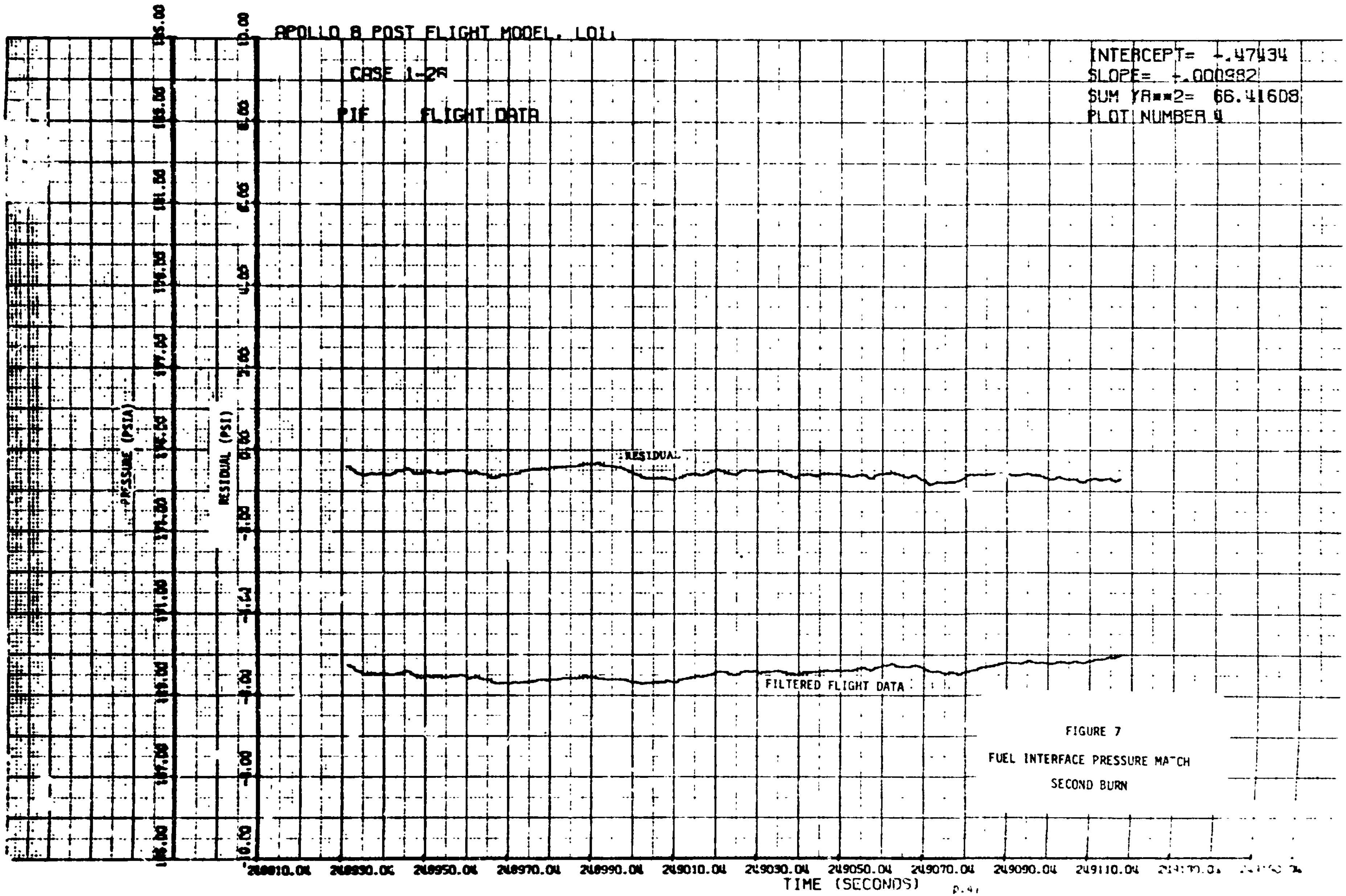


FIGURE 7

FUEL INTERFACE PRESSURE MATCH

SECOND BURN

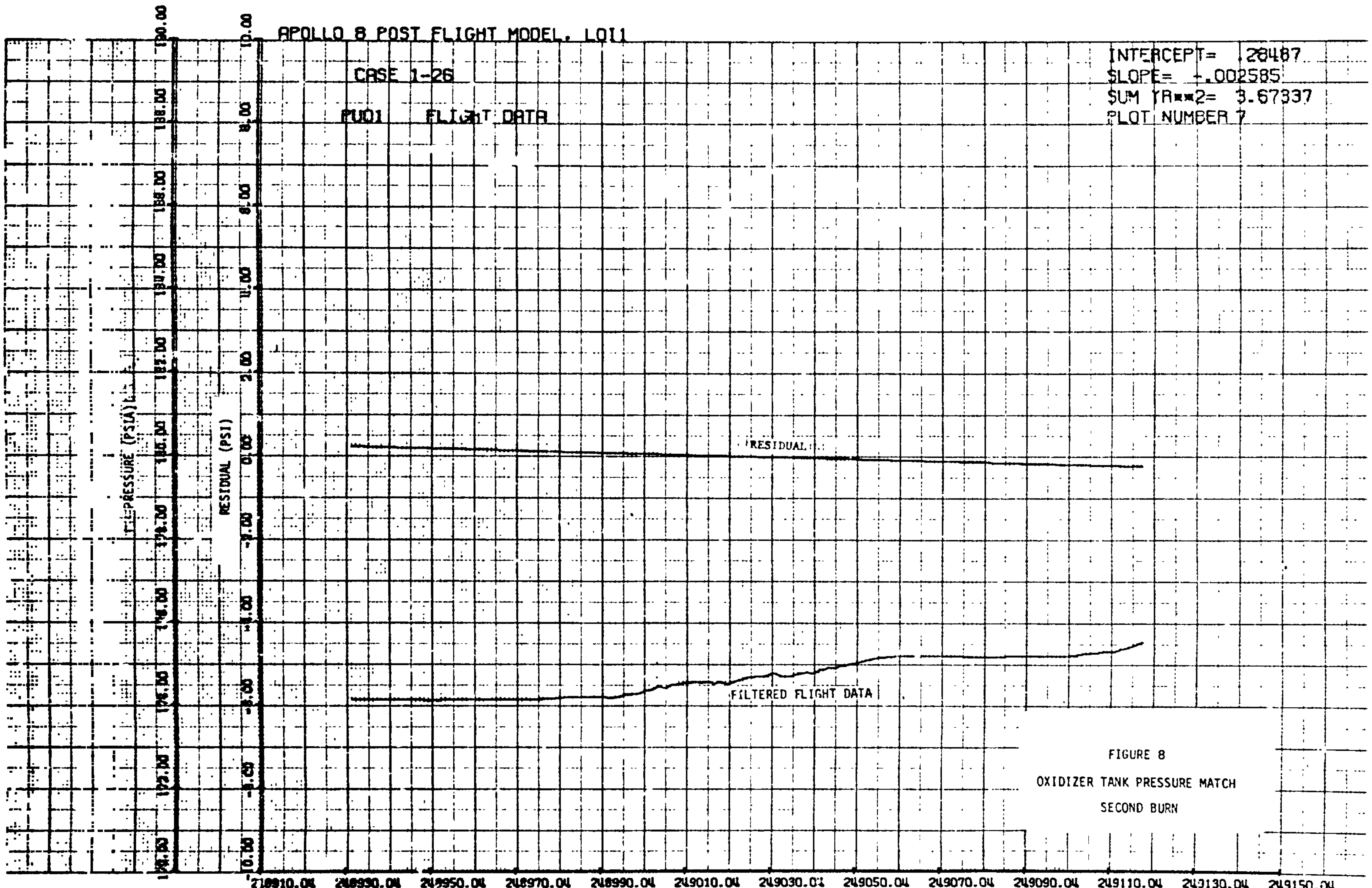


FIGURE 8

OXIDIZER TANK PRESSURE MATCH

SECOND BURN

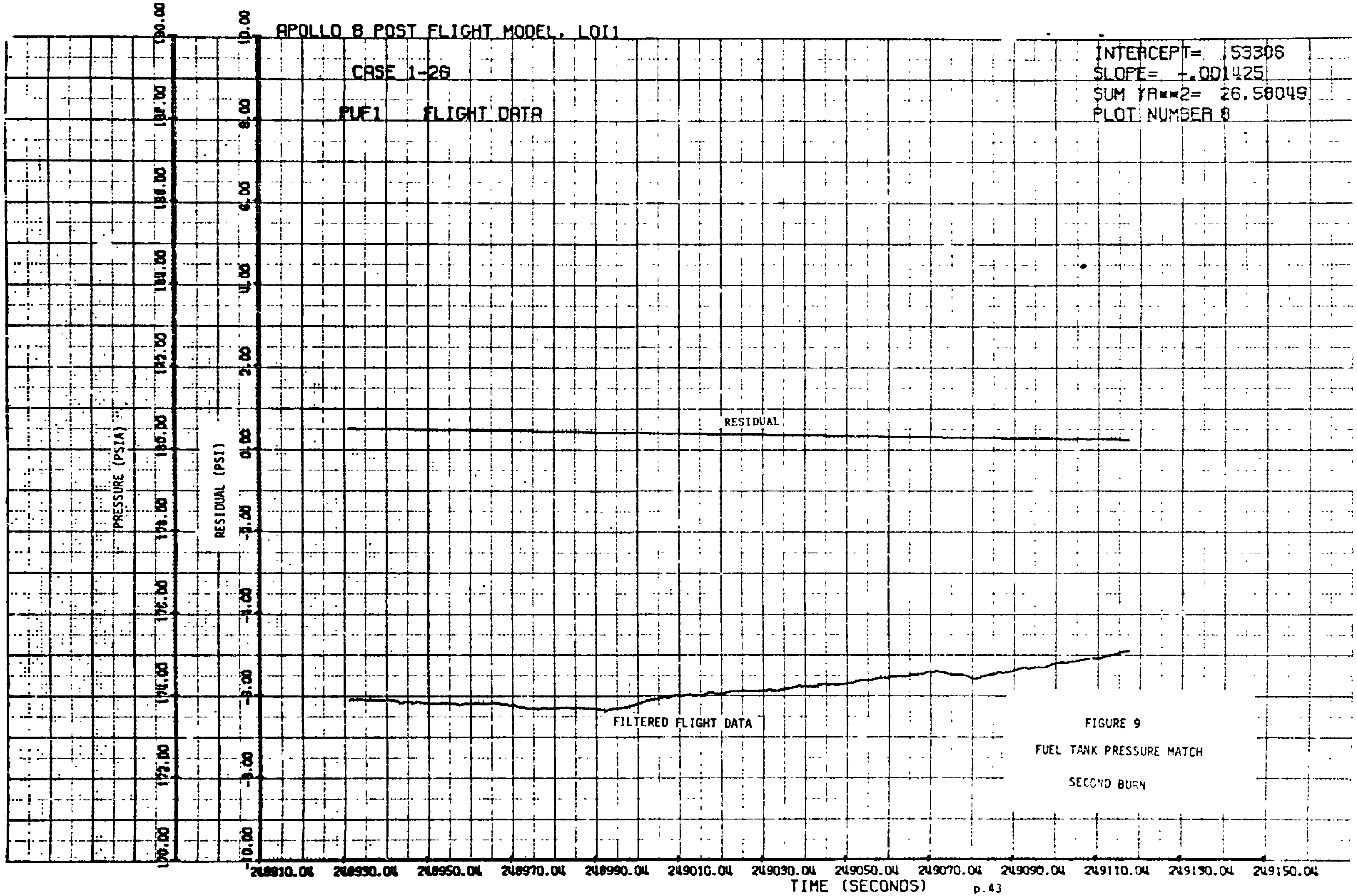


FIGURE 9

FUEL TANK PRESSURE MATCH

SECOND BURN

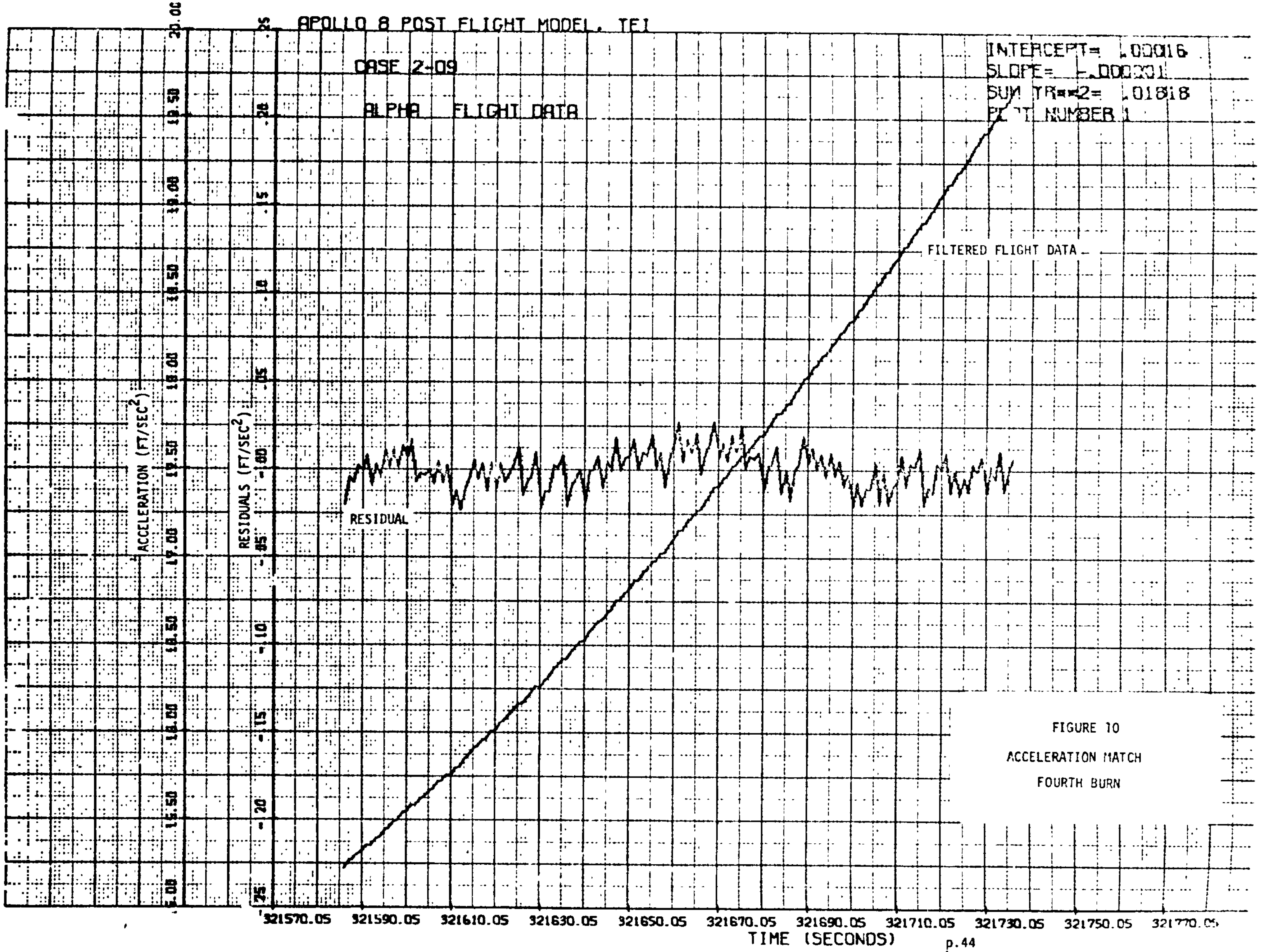


FIGURE 10

ACCELERATION MATCH

FOURTH BURN

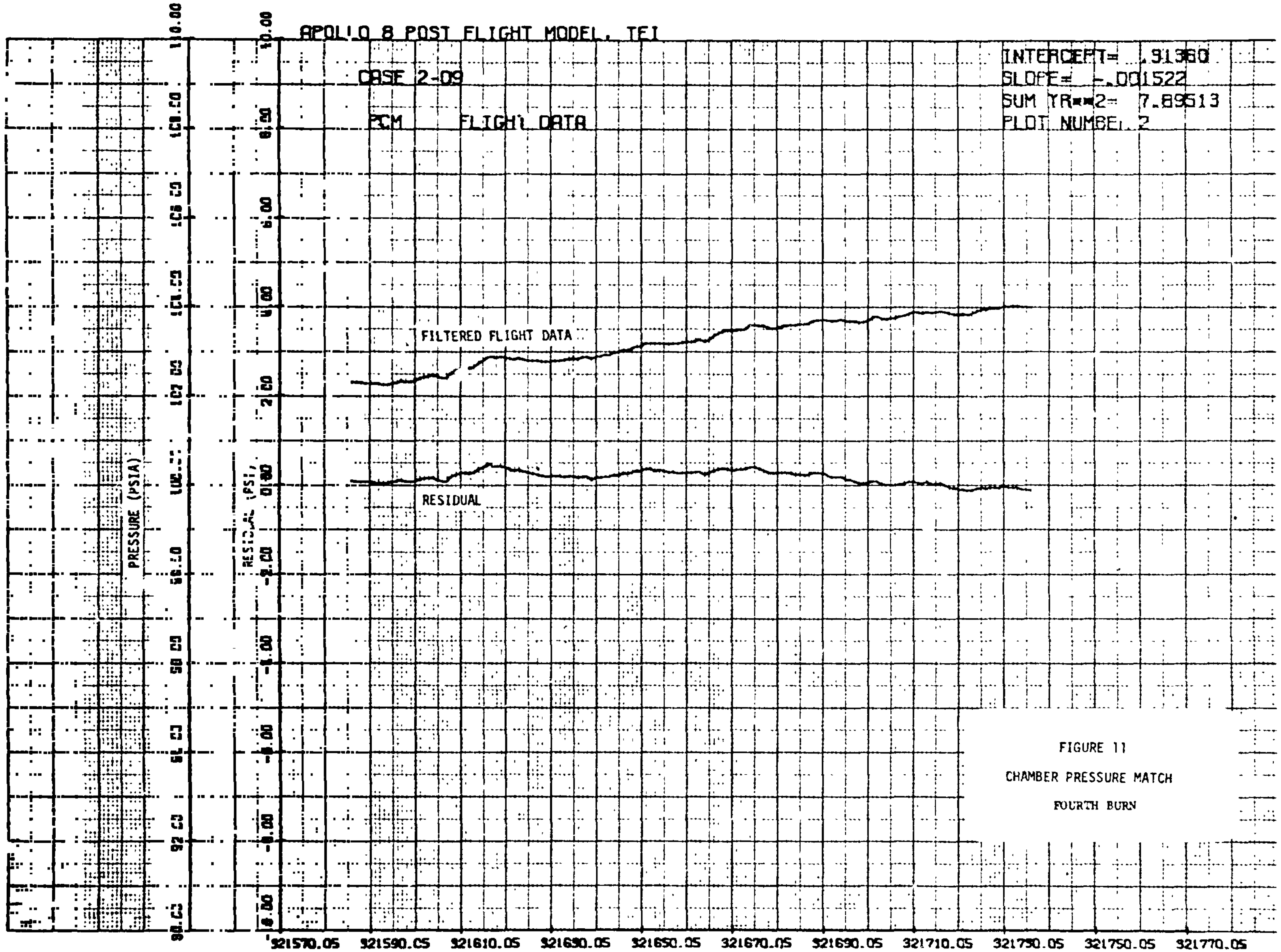


FIGURE 11

CHAMBER PRESSURE MATCH

FOURTH BURN

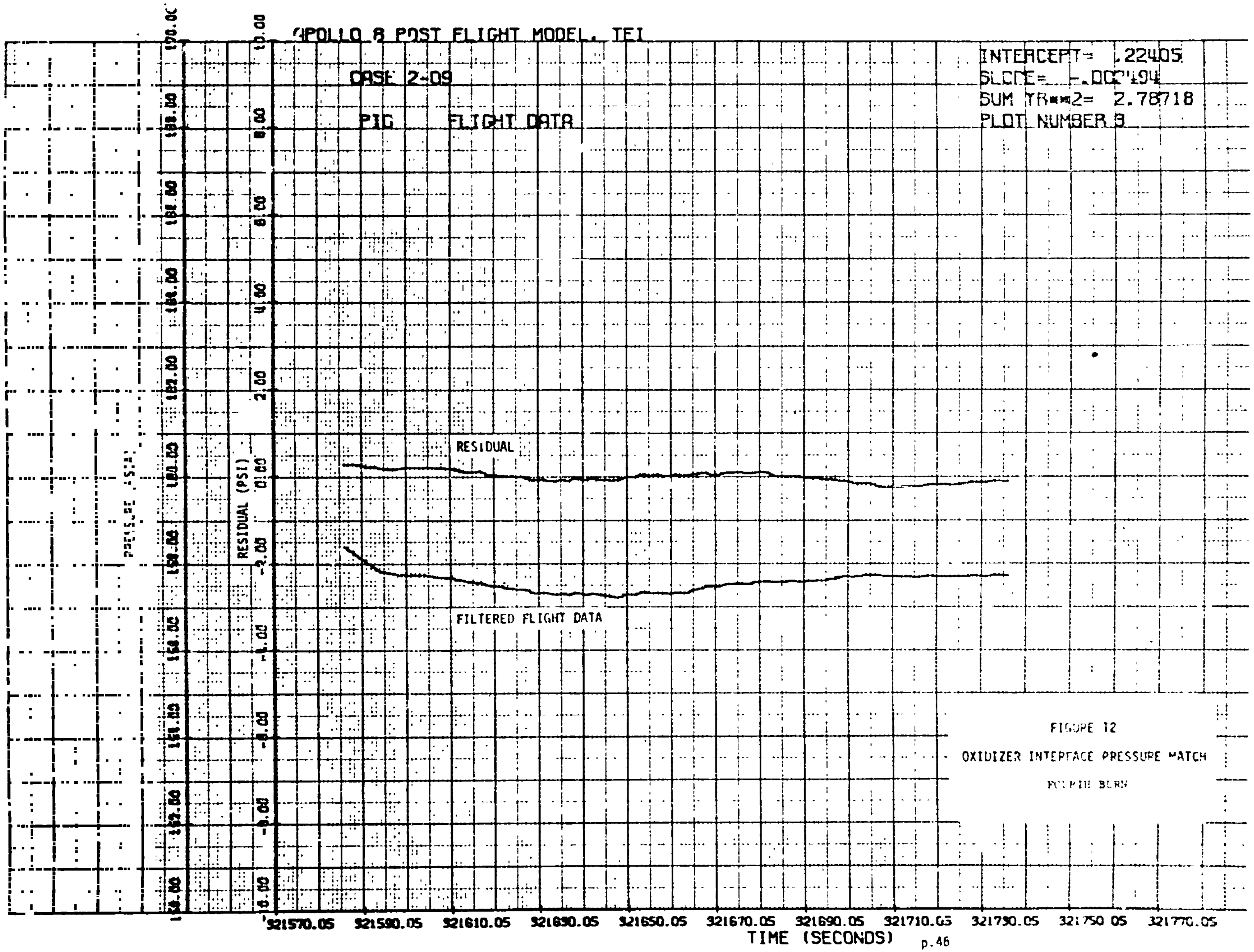


FIGURE 12
OXIDIZER INTERFACE PRESSURE MATCH

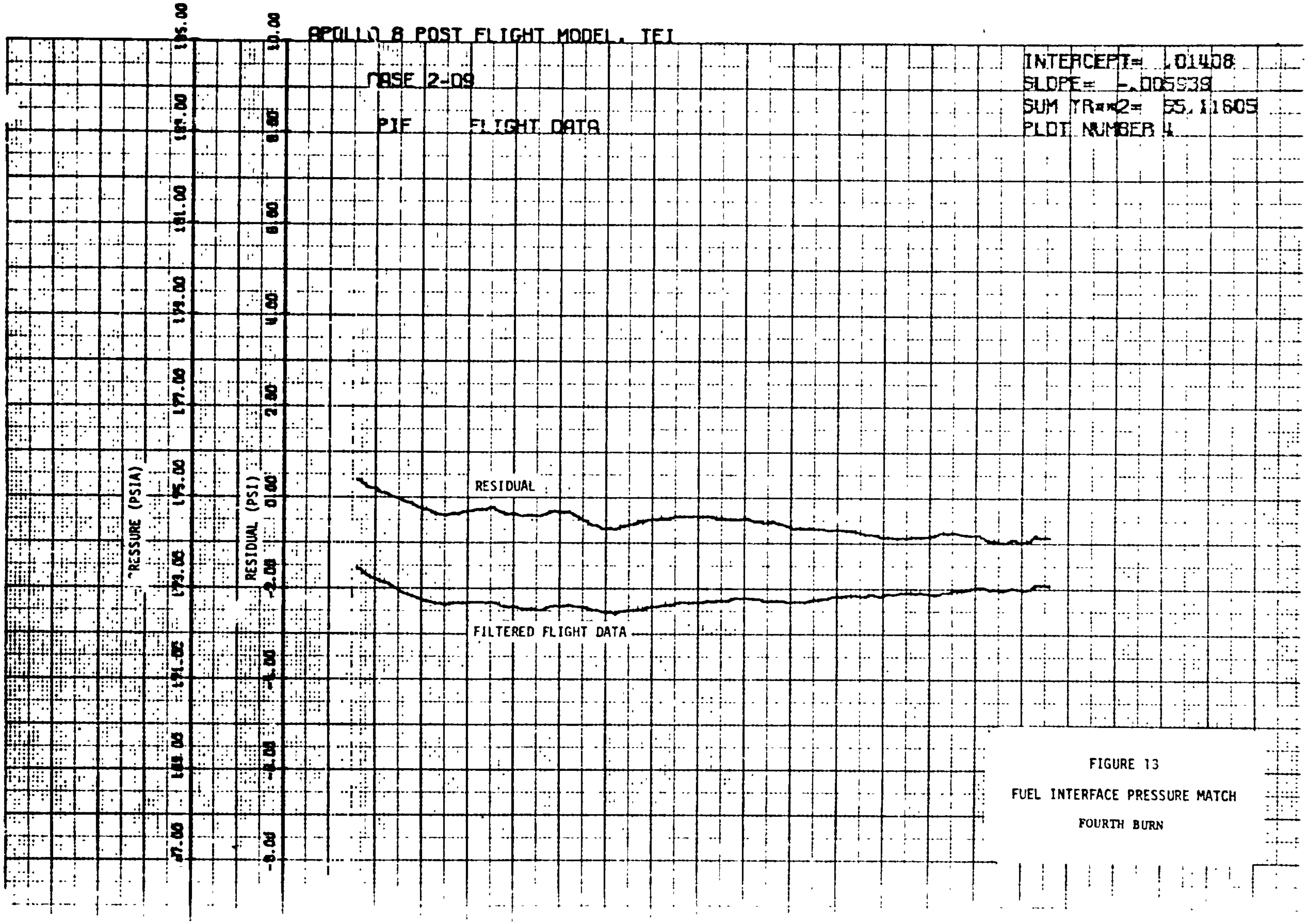


FIGURE 13

FUEL INTERFACE PRESSURE MATCH

FOURTH BURN

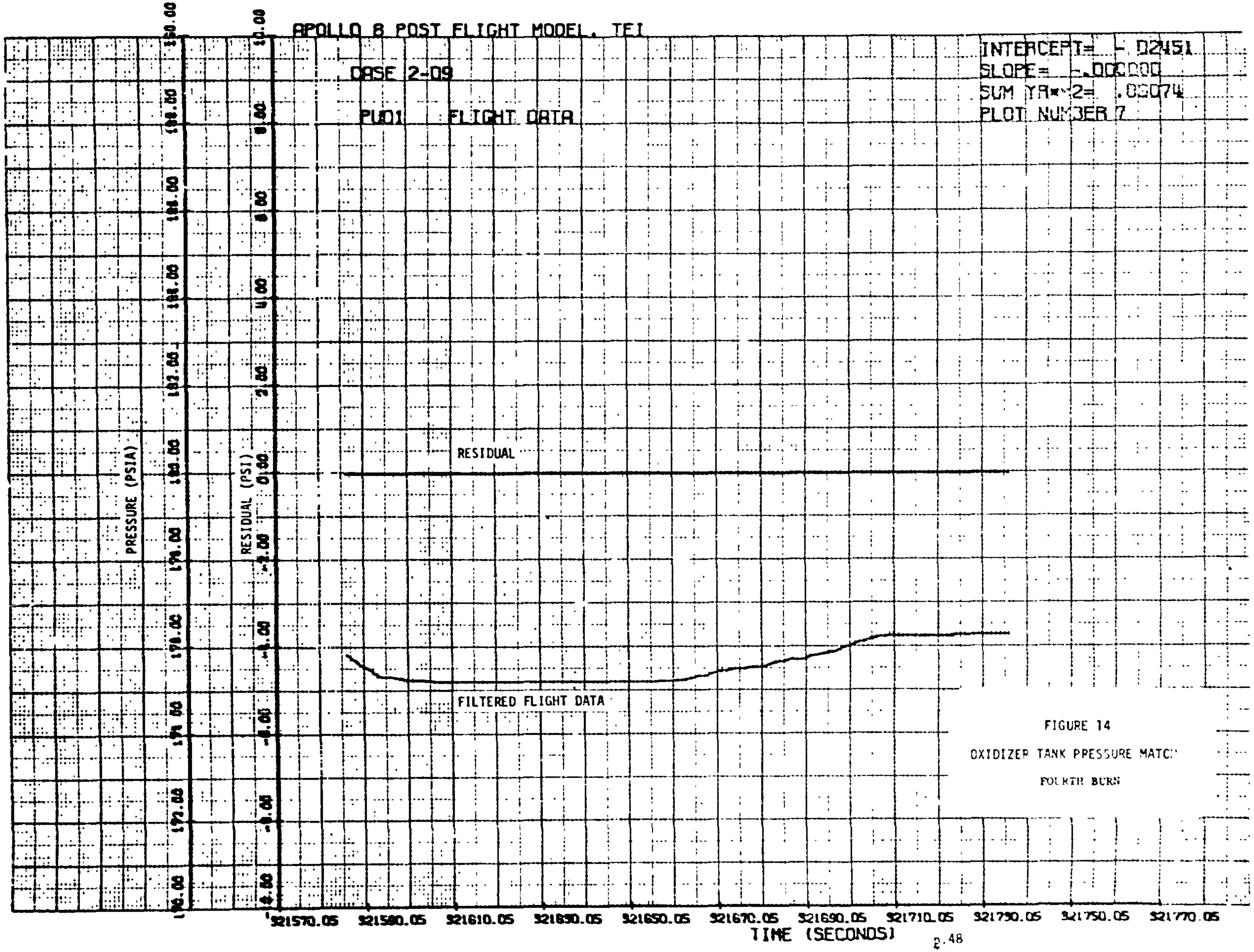


FIGURE 14

OXIDIZER TANK PRESSURE MATCH

FOURTH BURN

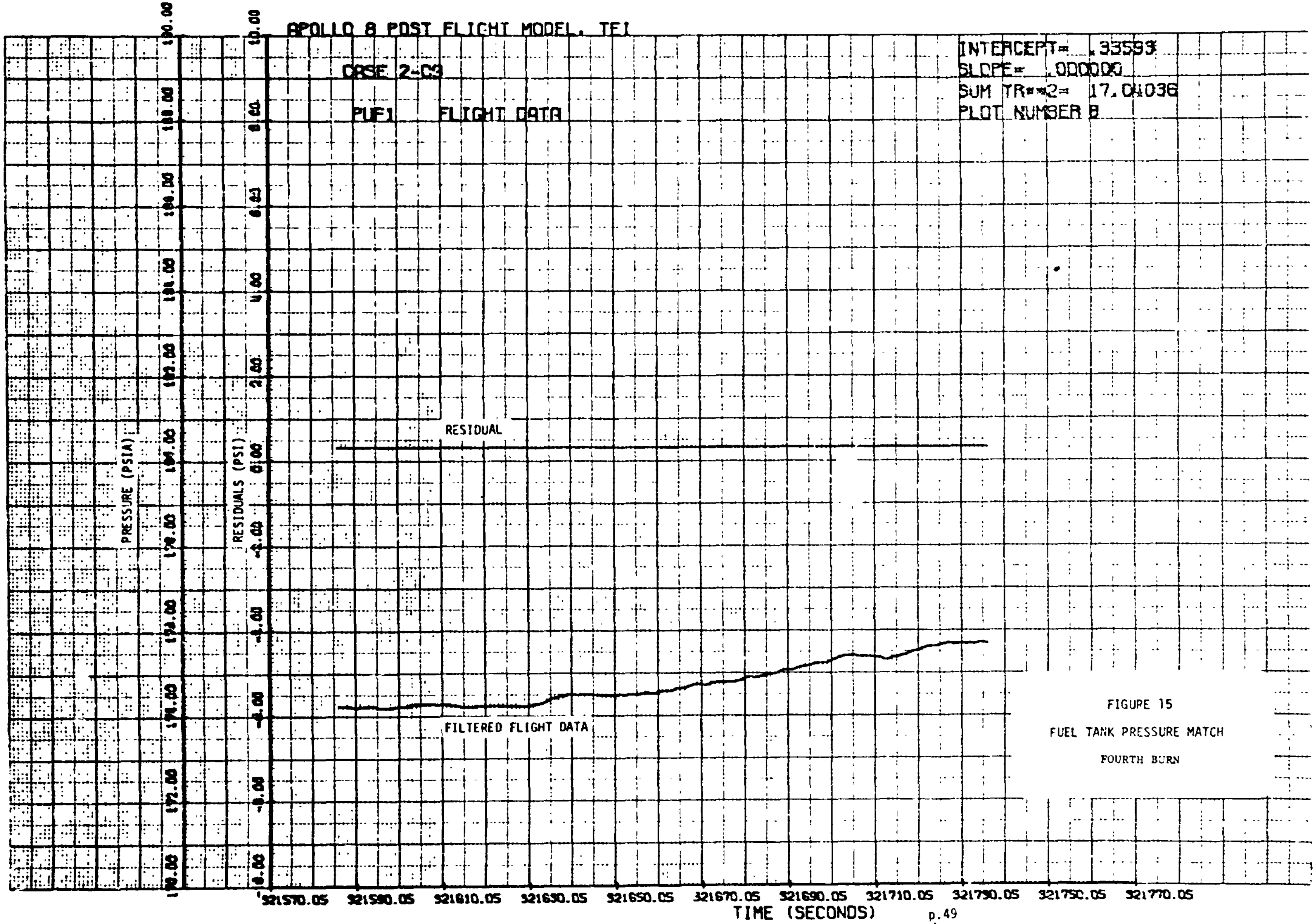
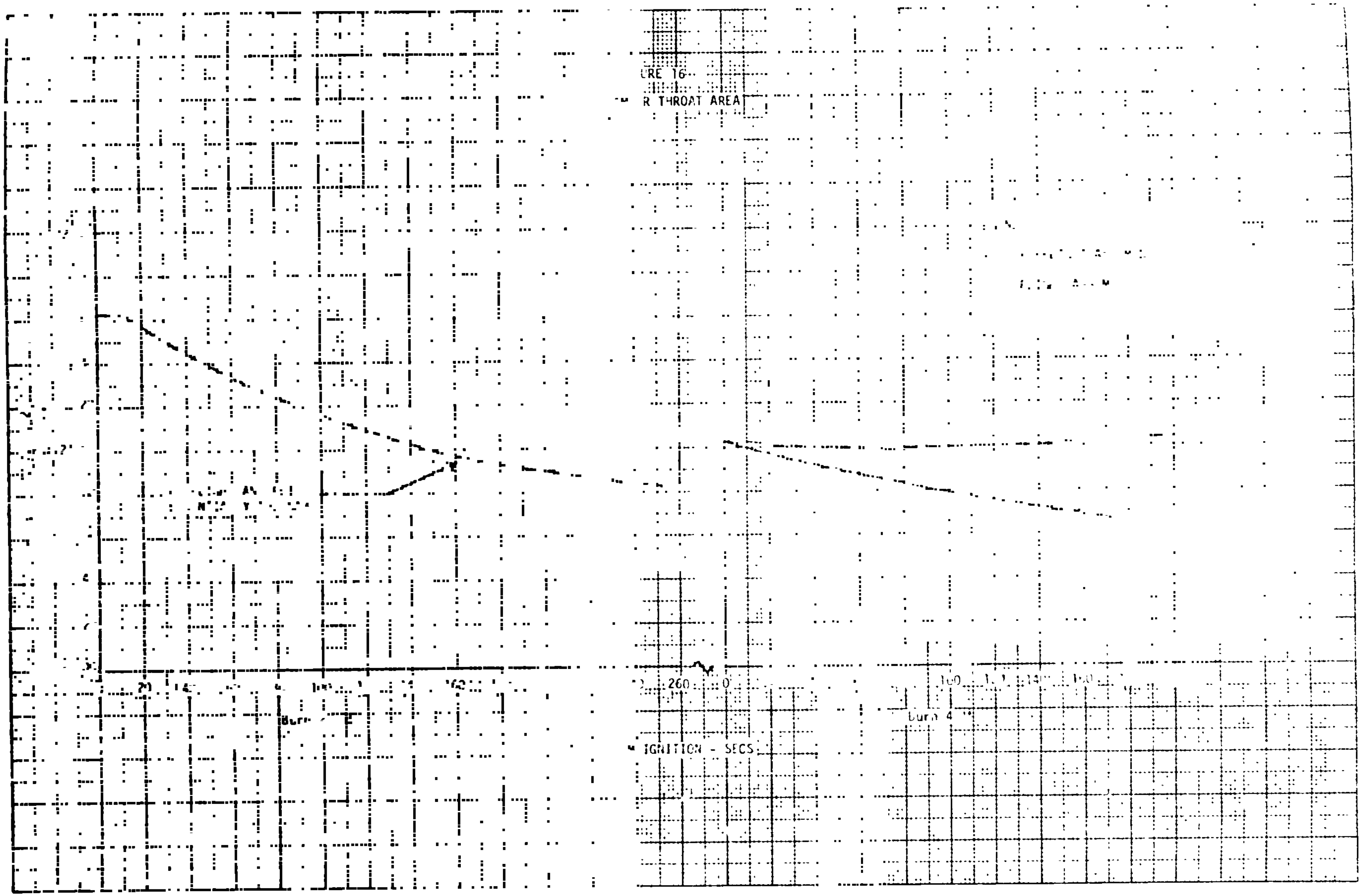


FIGURE 15

FUEL TANK PRESSURE MATCH

FOURTH BURN



URE 16
R THROAT AREA
260
0
Burn 4
IGNITION - SECS

FIGURE 17

COMPARISON OF PREFLIGHT PREDICTED AND INFLIGHT PERFORMANCE